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RESEARCH MEMORANDUM

PRESSURE DISTRIBUTION AND PRESSURE
DRAG FOR A HEMISPHERICAL NOSE AT MACH
NUMBERS 2.05, 2.54, AND 3.04

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

PRESSURE DISTRIBUTION AND PRESSURE DRAG FOR A HEMISPHERICAL NOSE AT MACH NUMBERS 2.05, 2.54, AND 3.04

By Leo T. Chauvin

SUMMARY

An experimental investigation of the pressure distributions on a hemispherical nose 3.98 inches in diameter, mounted on a cylindrical support, has been made at Mach numbers of 2.05, 2.54, and 3.04 and for Reynolds numbers of 4.44×10^6, 4.57×10^6, and 4.16×10^6, respectively. The Reynolds number was based on body diameter and free-stream conditions. Pressure-drag coefficients were calculated and good agreement was obtained between these tests and other investigations.

INTRODUCTION

From the standpoint of minimum drag, high fineness ratios and nearly pointed noses are desirable in the design of supersonic missiles and airplanes. However, it is necessary for some supersonic vehicles to have a hemispherical nose to house guidance equipment. Since use of this nose shape may result in severe drag penalties, with corresponding reduction in speed and range, the National Advisory Committee for Aeronautics has made tests in the preflight jet and in free flight to determine the pressure drag, pressure distribution, and aerodynamic heating of hemispherical nose bodies. The tests were made at the Langley Pilotless Aircraft Research Station at Wallops Island, Va. Flight tests to determine the drag of several round-nosed bodies at supersonic speeds have been reported in reference 1. The present paper presents the data obtained from pressure measurements at Mach numbers of 2.05, 2.54, and 3.04 and for Reynolds numbers of 4.44×10^6, 4.57×10^6, and 4.16×10^6, respectively. The Reynolds number was based on body diameter and free-stream condition.

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SYMBOLS

M	Mach number
C_p	pressure coefficient, $\dfrac{p - p_0}{\frac{1}{2}\rho_0 V_0^{\,2}}$
p	static pressure on body, lb/sq in. abs
p_0	free-stream static pressure, lb/sq in. abs
V_0	free-stream velocity, ft/sec
C_{D_W}	pressure-drag coefficient
H_0	stagnation pressure, lb/sq in. abs
ρ_0	free-stream density, slugs/cu ft
R	Reynolds number, based on body diameter
x	coordinate in free-stream direction
r	radius of hemispherical nose

TEST EQUIPMENT

The tests were made in the 8-inch auxiliary jet of the preflight jet of the Langley Pilotless Aircraft Research Station at Wallops Island, Va. Air for the operation of the preflight jet was stored in two spheres at 220 pounds per square inch absolute. A hydraulically controlled valve regulated the air from the sphere. The air then passed through a heat exchanger where it was heated sufficiently to compensate for the adiabatic temperature drop through the supersonic nozzle. The air next passed through a three-dimensional nozzle and exhausted to the atmosphere. Different test Mach numbers were provided by interchangeable nozzles. A shadowgraph system was provided for flow observations. Further information concerning the preflight jet can be found in reference 2.

The model was a hemispherical nose 3.98 inches in diameter, mounted on a cylindrical support, alined with the center line of the jet. Fourteen pressure orifices were located on the nose and spiraled rearward of the stagnation point to 0.6 inch on the cylindrical portion of the body. The ratio of model diameter to jet diameter was 0.498 for the $M = 2.05$ and $M = 2.54$ tests and 0.532 for the $M = 3.04$ test. Figure 1 shows the general arrangement of the nose mounted in the 8-inch auxiliary jet.

The test nose was made from spun K-Monel 0.05 inch thick, with a smooth and highly polished surface.

Measurements of the free-stream total pressure, free-stream stagnation temperature, and pressures on the body were obtained by a six-cell recording manometer, thermocouples, and electrical pressure pickups, respectively. These data were recorded by using two recording galvanometers and one optical recorder. The three recorders were synchronized with a timer of 10 cycles per second. The instruments used were accurate to 1 percent of their full scale.

The orifices on the model were 0.04-inch inside diameter, and the data presented were taken when the pressures on the body had reached a steady condition.

TEST CONDITIONS

The Reynolds numbers for the Mach numbers 2.05, 2.54, and 3.04 were 4.44×10^6, 4.57×10^6, and 4.16×10^6, respectively. The relatively constant Reynolds number of the tests was due to the variation of static pressure and temperature. The test at $M = 2.05$ was for a sea-level condition. The conditions for the tests at $M = 2.54$ and $M = 3.04$ represented a static pressure corresponding to altitudes of 12,000 and 28,000 feet, respectively.

RESULTS

The test results for Mach numbers 2.05, 2.54, and 3.04 are shown in figure 2 plotted as pressure coefficient as a function of the nondimensional parameter x/r, where x is the distance in the stream direction from the stagnation point on the nose to the measurement station and r is the radius of the hemisphere. Figure 2(a) shows the results for two tests made at $M = 2.05$ and at approximately the same free-stream total pressure. This figure shows that the pressure measurements can be repeated with good accuracy and that no duplication of the tests is needed.

Figure 3 shows a shadowgram taken for each test. Shocks eminating from the edge of the nozzle intersect the bow wave and influence the flow somewhere on the body. If the intersection was between the stagnation point of the body and the sonic point of the body, the pressure over the entire nose would be affected. For each Mach number tested the sonic point was estimated, and, in each case, the nozzle shock was found to lie well outside the subsonic region. For the $M = 2.05$ and $M = 2.54$ tests, the disturbance from the intersection does not affect the flow over the hemisphere. For the $M = 3.04$ case, however, the disturbance from intersection was stronger and caused a standing shock to occur on the hemisphere at a value of x/r of approximately 0.71 (see fig. 3(c)). Rearward of this point the measured pressures were affected by this disturbance and no data were presented for a value of x/r greater than 0.66. For the purpose of calculating the pressure drag for $M = 3.04$, the data were extrapolated to $\frac{x}{r} = 1.0$, as shown in figure 3(c).

Figure 4 shows the faired curves through the points of pressure coefficients plotted as a function of x/r for $M = 2.05$ of this series of tests. Also shown are the results from references 3 and 4 taken at $M = 1.90$ and $M = 1.62$ and at Reynolds numbers of approximately 0.44×10^6 and 0.811×10^6, respectively. The data from reference 3 were obtained with a hemispherical nose attached to a cylindrical body, and the data from reference 4 were calculated from interferometer observations on a sting-supported sphere.

The pressure-drag coefficient, plotted as a function of Mach number (fig. 5), was calculated by integrating the pressures from the present tests and from references 3 and 4. Pressure-drag coefficients obtained from reference 5 are shown in figure 5 for Mach numbers of 1.5, 1.98, and 3.02. The data from the references 3 to 5 correlate well with the faired curve through the points calculated from these tests.

CONCLUDING REMARKS

Pressure distributions were measured on a hemispherical nose 3.98 inches in diameter at Mach numbers of 2.05, 2.54, and 3.04 and for Reynolds numbers of 4.44×10^6, 4.57×10^6, and 4.16×10^6, respectively. The Reynolds number for these tests was based on body diameter and free-stream conditions. The pressure-drag coefficients obtained from previous investigations are in agreement with a faired curve through the points of pressure-drag coefficient calculated from these tests.

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National Advisory Committee for Aeronautics,
Langley Field, Va.

REFERENCES

1. Hart, Roger G.: Flight Investigation of the Drag of Round-Nosed Bodies of Revolution at Mach Numbers From 0.6 to 1.5 Using Rocket-Propelled Test Vehicles. NACA RM L51E25, 1951.

2. Faget, Maxime A., Watson, Raymond S., and Bartlett, Walter A., Jr.: Free-Jet Tests of a 6.5-Inch-Diameter Ram-Jet Engine at Mach Numbers of 1.81 and 2.00. NACA RM L50L06, 1951.

3. Moeckel, W. E.: Experimental Investigation of Supersonic Flow With Detached Shock Waves for Mach Numbers Between 1.8 and 2.9. NACA RM E50D05, 1950.

4. Wood, George P., and Gooderum, Paul B.: Method of Determining Initial Tangents of Contours of Flow Variables Behind a Curved, Axially Symmetrical Shock Wave. NACA TN 2411, 1951.

5. Sommer, Simon C., and Stark, James A.: The Effect of Bluntness on the Drag of Spherical-Tipped Truncated Cones of Fineness Ratio 3 at Mach Numbers 1.2 to 7.4. NACA RM A52B13, 1952.

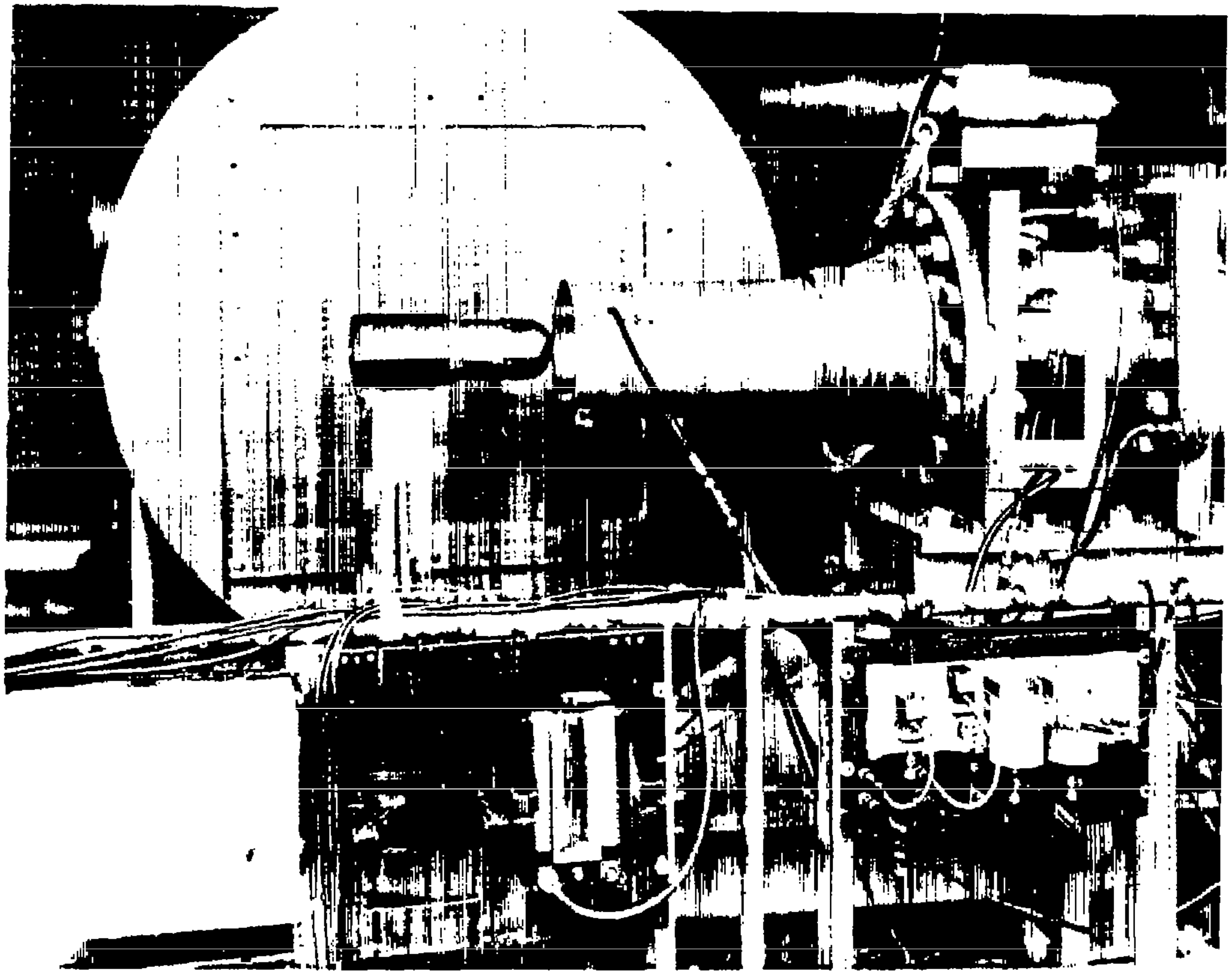
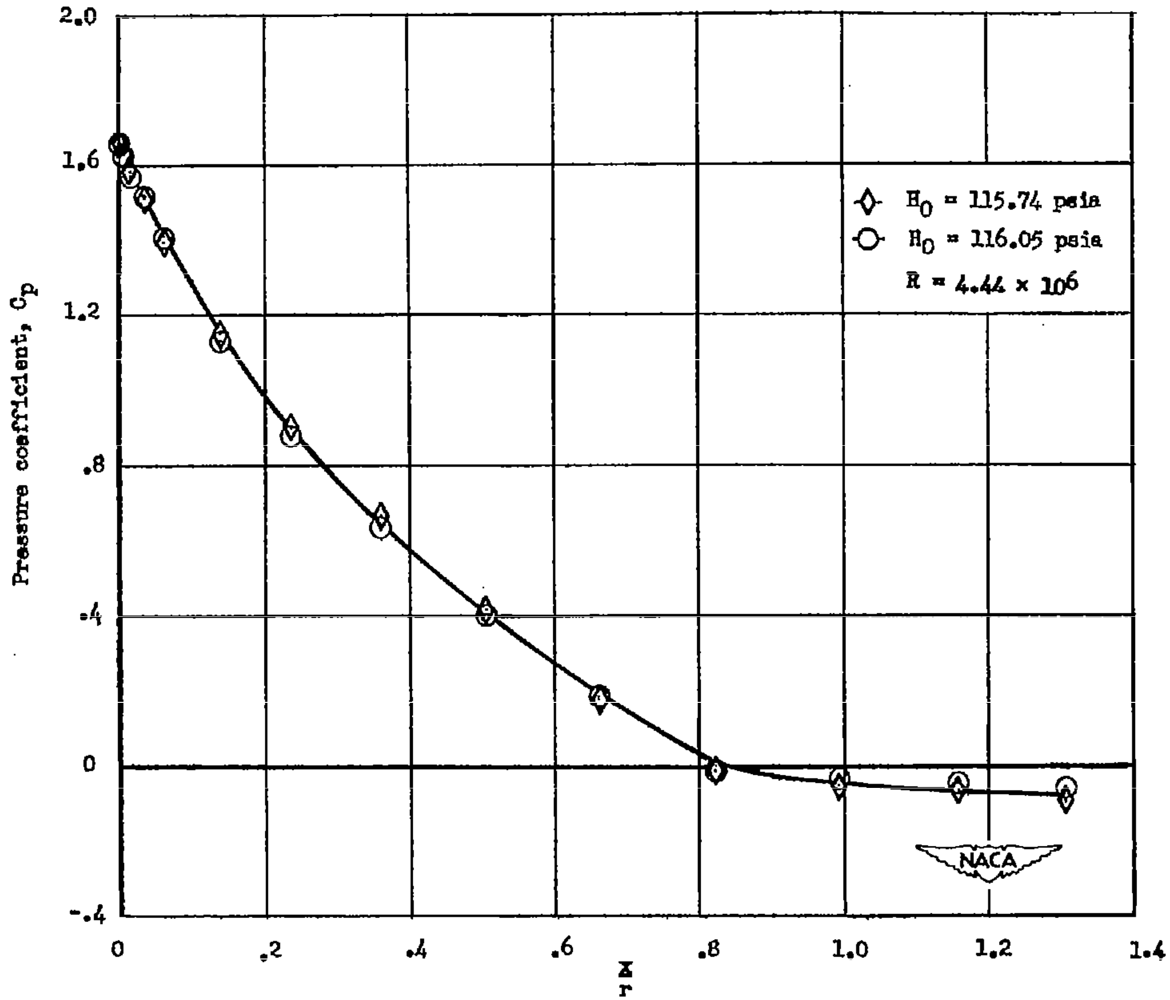
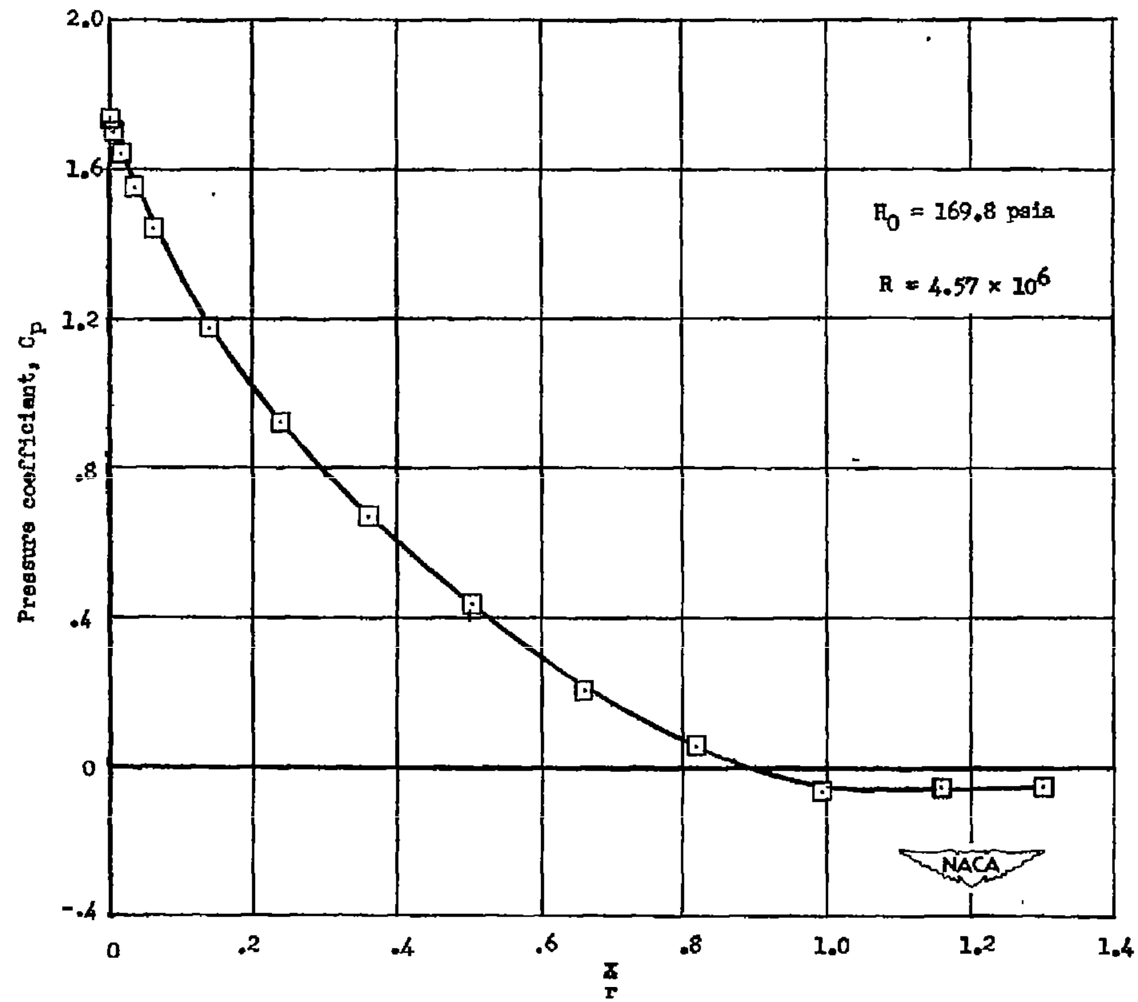


Figure 1.- Photograph of model mounted in the 8-inch auxiliary jet. L-74485



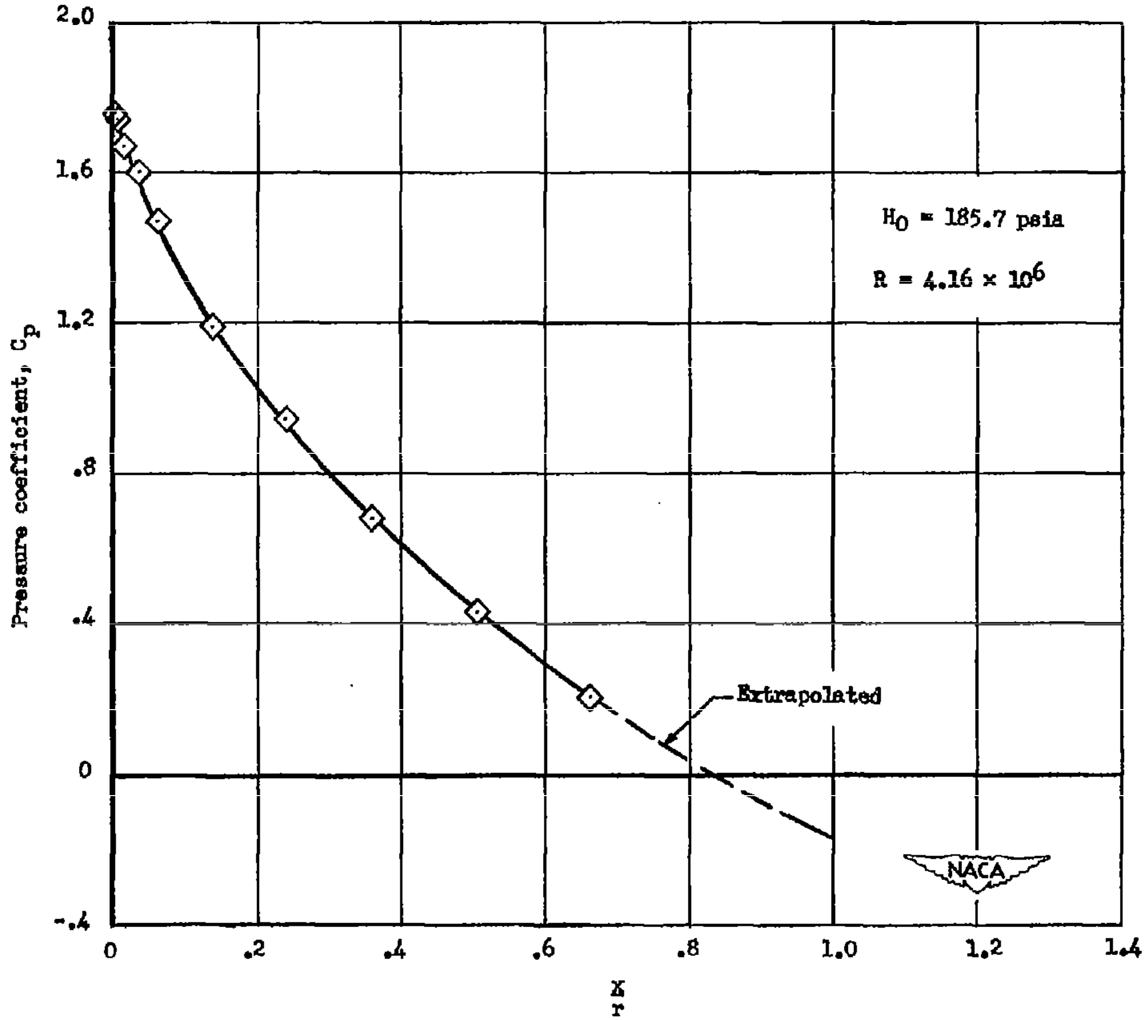
(a) M = 2.05.

Figure 2.- Pressure distribution on a hemispherical nose.



(b) M = 2.54.

Figure 2.- Continued.



(c) M = 3.04.

Figure 2.- Concluded.

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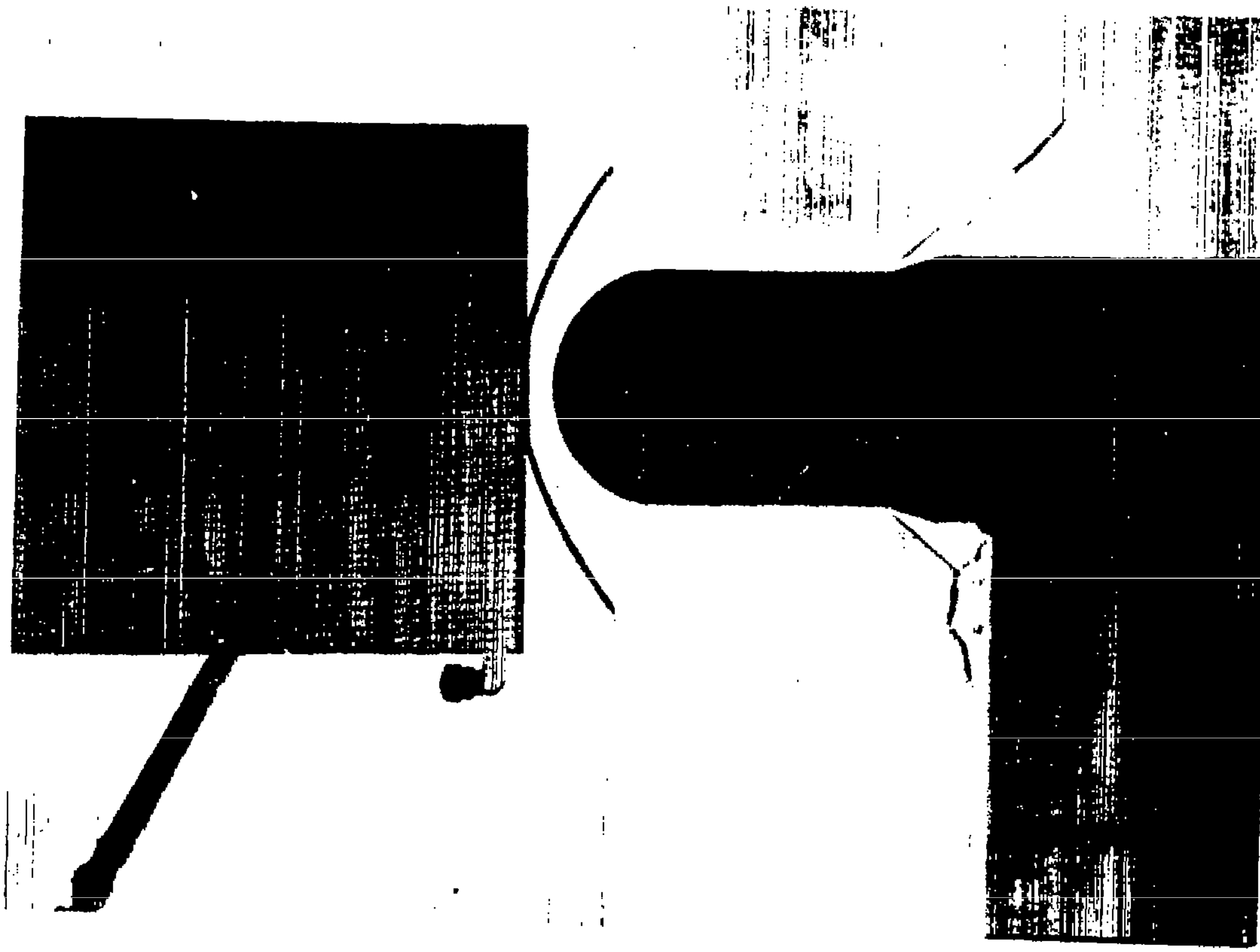

(a) M = 2.05.

Figure 3.- Shadowgram for each test. L-74787.1

(b) M = 2.54.

Figure 3.- Continued. L-74788.1

(c) $M = 3.04$.

Figure 3.- Concluded.

L-74789.1

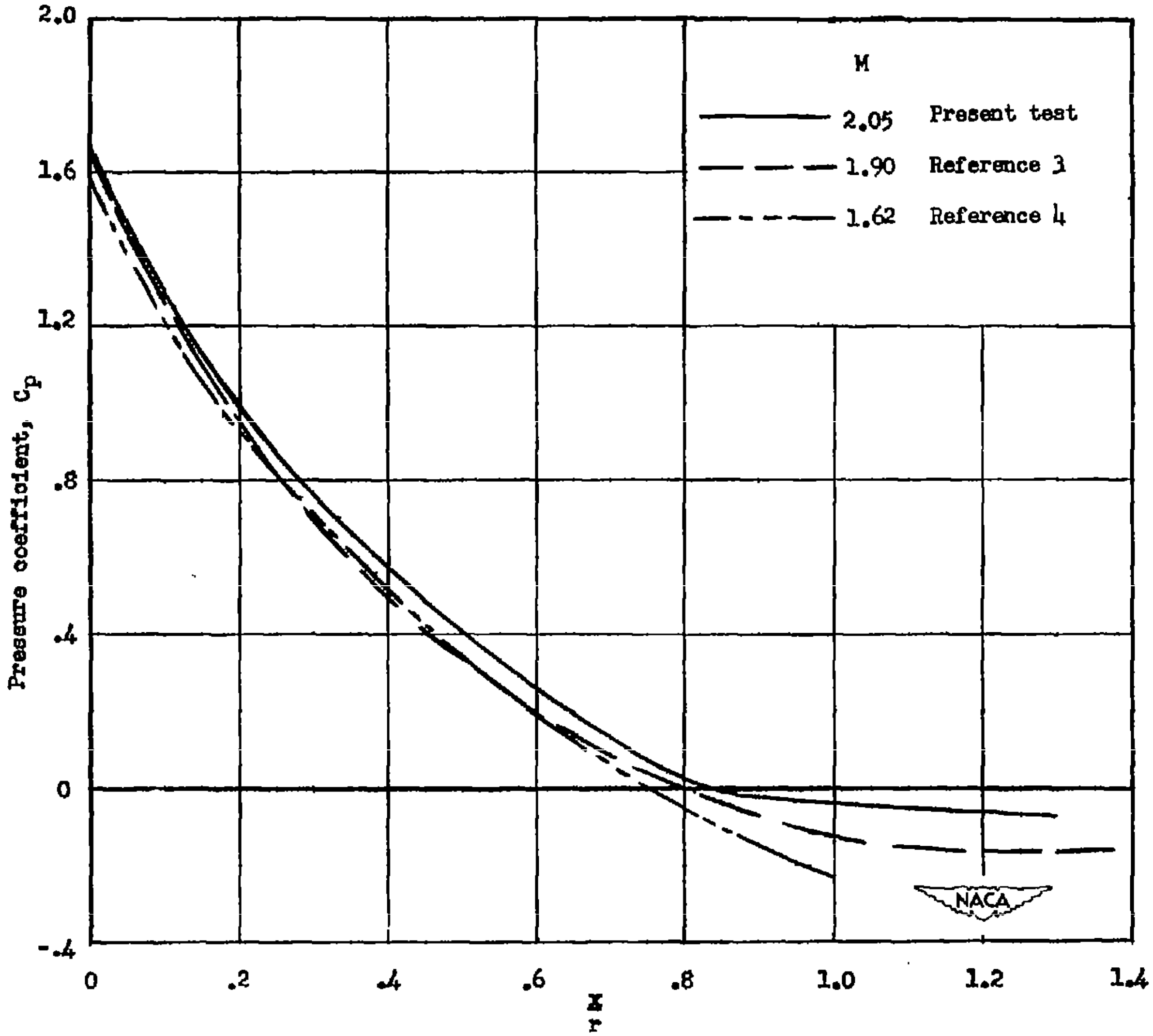

Figure 4.- Pressure distribution on a hemisphere for M = 2.05 test and references 3 and 4.

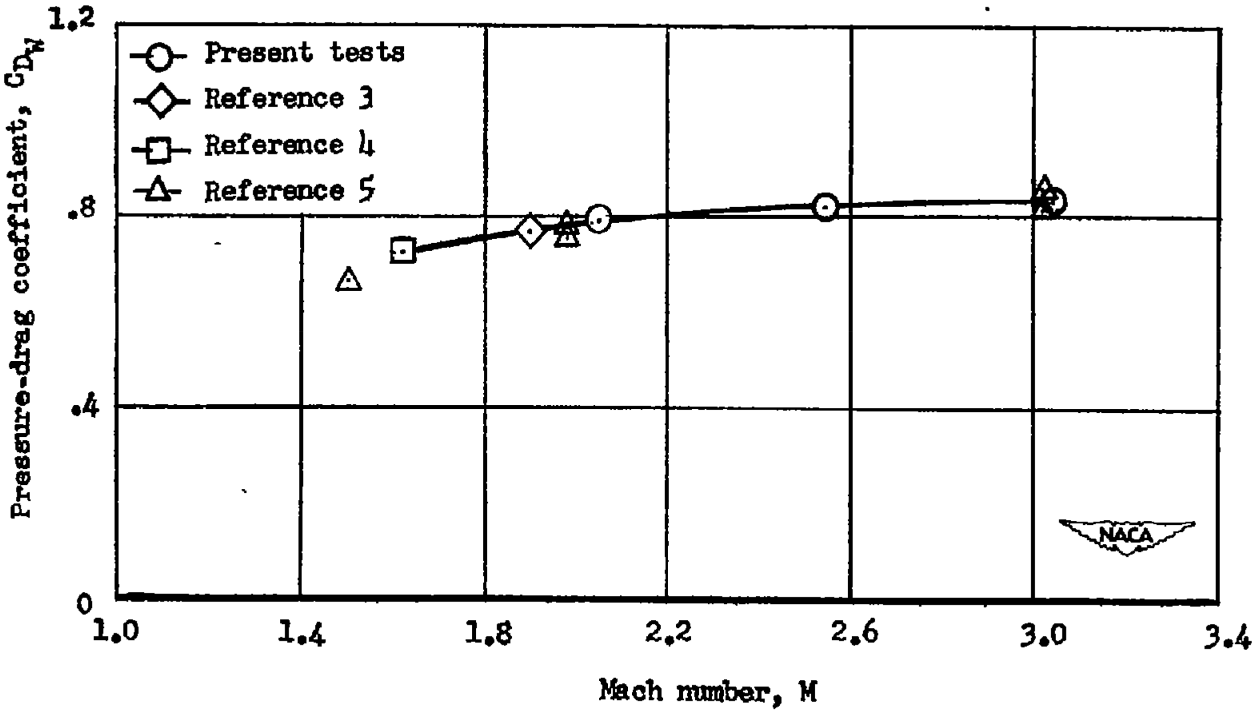

Figure 5.- Variation of pressure-drag coefficient with Mach number for a hemisphere.

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